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PROSPECTS FOR A MULTIPURPOSE SOLAR-ELECTRIC PROPULSION STAGE

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PREFACE

The present memorandum records several aspects of a preliminary solar-electric multimission applications survey and status review conducted by the Lewis Mission Analysis Branch in mid-1970. This information is presented in the interest of communication, and because in some respects, it may either confirm or offer alternatives to other studies in progress.

The reader is cautioned that the word "stage" is used here in its broadest sense. That is, a solar-electric "stage" is considered by the authors to include everything that is separated from the basic Earth-launch vehicle except the science packages. By its nature, the electric stage must incorporate many functions such as long-term attitude control and guidance that are now found in a traditional spacecraft such as Mariner. It also includes an electrical power source (the solar array system) that is probably adequate for data-gathering and telemetry purposes at the mission destinations. Intuitively, it would seem undesirable to duplicate these functions in a separate, independent spacecraft. The science packages therefore have "passenger" status on board a "stage" or "bus" which is capable of flying a complete mission trajectory by itself.

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ABSTRACT

A review of current solar-electric propulsion (SEP) technology is given along with preliminary performance estimates for a fixed design multipurpose SEP stage or "bus" over a broad spectrum of missions and two launch vehicles. Results for an arbitrarily chosen 10 kilowatt, 2600 second specific impulse design show that a single SEP vehicle would have excellent performance capability. Its performance margin over a TE 364-4 chemical stage is especially great for the more difficult missions. No technical problems are apparent now that might preclude the successful demonstration of flight-rated hardware by the latter 1970's.

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SUMMARY

The basic technology of solar-electric propulsion is in hand. This technology is now ready to be incorporated into the design of a single multipurpose spacecraft. Although the component weights and efficiencies are quire acceptable, endurance testing of a complete prototype system under realistic environmental conditions is still required.

SEP mission performance does not generally depend critically on the specific values of power level and specific impulse. Although the first cut values (10 kW and 2600 sec) used here to evaluate the performance potential should be refined, they lead to a very favorable performance evaluation. In comparison with a TE364-4 powered chemical upper stage, the SEP system provides either (1) equal payload when using a smaller launch vehicle, or (2) significantly greater payloads using the same launch vehicle. The SEP system can perform some very difficult missions such as the Mercury orbiter and asteroid rendezvous which are virtually impossible for the chemical system. SEP mission time for outer-planet missions is usually lower than that for chemical systems. But it is higher for most other missions, although in the case of area-type missions this is not judged to be a large penalty since useful data is generated continuously throughout the flight.

Two launch vehicles, Atlas/Centaur and Titan IIID/Centaur, are shown to accommodate a wide range of mission targets, flight times, and payload levels using a single SEP design. Thus it is suggested that an SEP spacecraft be physically and functionally compatible with the Centaur stage.

Especially attractive missions for early application of SEP from a technical standpoint are the 1 AU extra-ecliptic and solar monitor. The synchronous communications satellite and planetary missions are also noteworthy from the applications and scientific interest viewpoints; however,

INTRODUCTION

In little more than one decade, our unmanned space payload capability has increased from a few pounds in a low Earth orbit, to sending nearly a thousand pounds towards Mars and Venus. This dramatic improvement is due to the introduction of larger and more efficient launch vehicles such as Atlas/Centaur and the Titan family. These vehicles provide the capability to perform significant and rewarding missions such as near-planet flybys and low energy, area-type space probes. The missions themselves are generating a wealth of basic scientific data, and also engineering information that will help in planning more ambitious missions.

On the other hand, future programs are likely to emphasize larger payloads and higher launch energies. Present vehicles, despite their distinguished record, become inadequate at some point and one must then consider the alternatives.

One approach has been to simply add small solid upper stages to a standard launch vehicle such as Atlas/Centaur. This gives substantially better performance at moderate cost. Its great advantage for future planning purposes is that it is based on familiar, proven technology. Unfortunately, the performance obtained in this way is not really adequate for the most demanding missions of interest; and even for less difficult missions, it is often necessary to use the largest available booster to get a satisfactory payload.

Thus, to perform the more difficult missions, and to avoid the expense of developing new, larger and more sophisticated launch vehicles, it is reasonable to consider the alternative of applying advanced, high performance upper stages to existing boosters. One very attractive upper stage concept would rely on solar-electric power for primary propulsion. As illustrated in figure 1, the incident solar energy is first converted to electricity by large solar cell arrays, and is then routed to a set of electric thrusters. This system is of special interest for reasons of (1) availability, (2) suitability, and (3) performance:

1. As will later be described in detail, the necessary technology appears to be available or nearly so. The SERT-II experiment has already demonstrated the basic feasibility and flightworthiness of solar electric propulsion beyond dispute. Lewis in-house and contractor design studies, based on recent research findings, agree that major subsystems and components can now be built which have adequate performance and lifetimes.

2. Secondly, the solar arrays are not subject to a minimum-size restriction (as a nuclear reactor, for example, would be). Therefore they can be built efficiently in the modest sizes that are suitable for unmanned probe missions and compatible with available launch vehicles.

3. Finally (as will be shown) the potential solar-electric stage also offers very good performance over a broad spectrum of missions. In comparison with chemical upper stages, it offers greatly improved payload when the same booster is used. Alternatively, the solar-electric stage on a small booster performs comparably to a much larger all-chemical system for most missions.

The subject of solar-electric propulsion has been studied extensively since 1964. (Detailed and recent surveys of the major developments and literature contributions may be found in refs. 1, 2, and 3.) The first serious study of using a standard multipurpose solar-electric stage was published in late 1967 (ref. 4). This study, based on work by Meissinger and others at TRW, considered the performance potential of a simple, comparatively small solar-electric vehicle for solar probe, solar monitor, extra-ecliptic, and asteroid belt missions. Because of its low power (3 kW) and low acceleration, this particular vehicle required relatively long trip times to match the performance of an efficient chemical system. In 1968, Zola (ref. 5) looked forward to more demanding missions (major planet flybys and orbiters) and a much more sophisticated space vehicle. Using presumably conservative input assumptions, he found that a fixed design, 10 kW power, 4500 second I_{sp} stage could feasibly visit each of the major planets, with little performance penalty compared to continuously re-optimized designs. The same study was later extended to include Mars and Mercury orbiter missions (ref. 6). Comparisons showed that a hypothetical, very advanced (H-F) kick stage offers the best performance for the Mars and Jupiter missions, while the solar-electric stage is superior for the Mercury orbiter, and for the Uranus and Neptune flyby missions. The two systems appeared to be competitive for Saturn missions.

A more recent TRW study (ref. 7) included a relatively complete vehicle conceptual design effort in addition to performance estimates. A fixed vehicle with 6.4 kW of power and a 3200-second specific impulse was shown to have good performance for extra-ecliptic, asteroid belt, and Jupiter flyby missions. Performance comparisons for chemical and solar-electric upper stages were made for the Jupiter flyby mission. Realistic chemical-stage parameters (Burner II rather than Zola's hypothetical H-F stage) were used, and allowance was made for practical factors such as solar-cell degradation that tend to penalize the solar-electric stage's performance. On this basis of comparison, the solar-electric stage was able to deliver the same payload in a 33 percent shorter Earth-Jupiter travel time than the all-chemical system (400 vs. 600 days).

Studies of a synchronous satellite mission by Hrach (ref. 8) and Reader and Regetz (ref. 9) made the point that a solar-electric stage on a small booster can often match the performance of a considerably larger all-chemical system. The implied launch-cost savings could help to underwrite development of a solar-electric stage, especially one that has multimission capability. It is of interest to note that references 8 and 9 recommend values of power and specific impulse which were not greatly different from Zola's 10 kW and 4500 seconds. Thus, they contribute to the case for a standard solar-electric stage even though they did not specifically consider that possibility. The same is true of several other single-mission studies. For example, Strack and Hrach considered extra-ecliptic missions in reference 10. They concluded that a fixed stage with 10 kW of power and 2600 seconds specific impulse can greatly exceed the performance of competitive chemical systems, and would be suitable for a wide range of final inclinations, mission times and launch vehicles.

The contributions outlined above (together with many others not mentioned) collectively amount to a strong a priori case for some kind of a solar-electric space vehicle. But, because there has been little uniformity among the various studies in mission objectives, technology inputs, and launch vehicles used, several major questions have been unanswered.

1. What are the most appropriate values of power, specific impulse, and other major design parameters for a fixed-design multipurpose solar-electric stage?

2. What levels of component and system technology are necessary to give attractive performance and when might these be expected to be available?

3. For which missions and launch vehicles will this stage perform effectively?

4. Can the unique characteristics of electric propulsion (i.e., guidance, attitude control, electrical power and many "housekeeping" functions are available on a long-term basis)

be translated into payload simplifications or enhanced operational capabilities?

First-cut answers are proposed in this paper. The next section reviews the effect of power and specific impulse upon the performance and operating characteristics of a solar-electric stage and makes a tentative choice of those parameters. The technology and design-point values thus defined are used as input data for performance estimates. A wide range of mission destinations, objectives, and launch vehicles are studied on a consistent basis. Performance comparisons with an efficient solid-propellant chemical upper stage (TE364-4) are included in each case.

The "Propulsion Technology" section then provides a review and current status summary of thrusters, solar cell arrays and related technology items. Its object is to explain and justify the input values used for the mission calculations and to identify potential problem areas.

MISSION PERFORMANCE

The basic propulsion-related technology values used in this study are presented in the following list with justifications and detailed discussion deferred to the "Propulsion Technology" section.

Total propulsion system specific weight ( )	30 kg/kW
Propellant tankage factor (percent of propellant mass)	0.10
Thruster efficiency at I_{sp} = 2600 sec.	0.625
Power conditioning efficiency	0.91

For the purposes of this section, it is sufficient to state that these values are felt to be achievable through the normal development of currently available laboratory technology even though the individual compoenents have not yet attained a flightworthy status.

Propulsion System Design Parameters

Based on these values, a single, standardized solar-electric upper stage is felt to provide a very attractive multimission capability. To demonstrate this, consider the mission performance of the 10-kW, 2600-second specific impulse solar-electric stage that emerged from the recent out-of-the-ecliptic mission study by Strack and Hrach (ref. 10). This choice is admittedly arbitrary and is intended only as the first step in an iterative cycle. On the other hand, it is not completely without foundation. First, recall that 10 kW and 2600 seconds were recommended by Strack and Hrach as good compromise values that would be satisfactory for out-of-ecliptic mission with a wide variety of final inclinations, mission times, and launch vehicles; i.e., these values were already considered to represent a multimission vehicle within the context of reference 10. Secondly, it should be understood that the payload maxima associated with optimum power and I_{sp} are usually very broad and flat. Thus, it is often possible to depart substantially from "optimum" conditions without incurring unacceptable penalties. This is demonstrated in figure 2, which illustrates how payload varies with these two variables. Curve A is for a 150-day synchronous satellite mission using an Atlas/Centaur launch vehicle. In this case, the payload drops about 30 percent if the power is reduced from 20 to 10 kilowatts. But the other two curves are quite flat, which is more typical behavior. Curve B is for a 700-day rendezvous with the asteroid Ceres, again using an Atlas/Centaur launch vehicle. Curve C is for a 500-day 0.1 AU solar probe and uses a Titan IIIC booster. Because the solar cell arrays are quite expensive -- e.g., $500,000 per kW -- it is not sufficient to consider payload performance alone. As a matter of judgment, we feel that the performance versus cost tradeoff will optimize somewhere near the 10 kilowatt level shown with the dotted line. It can be seen from the other set of three curves, that payload is also rather insensitive to specific impulse. Values between 2000 and 5000 seconds all would apparently give satisfactory performance. In addition to these results, it may be recalled that similar values have emerged from other studies. For example Reader and Regetz arrived at a power of 9 kW and 3000 seconds specific impulse in their study of a Delta boosted synchronous communications satellite raising mission (ref. 9). As previously mentioned, Zola (ref. 5) proposed 10 kW and 4500 seconds for outer planet flyby missions and orbiters of Mercury and Mars.

In view of these points there is every reason a priori to believe that a stage with 10 kW of power and 2600 seconds specific impulse would give satisfactory performance for a wide range of mission and launch vehicles, even though it is optimum for none of them. To demonstrate the point, these values will be taken as representative of a fixed design solar-electric propulsion stage (henceforth identified by the initials, SEP) in the remainder of this paper. The power of 10 kW requires a pair of (approximately) 10-foot by 50-foot solar arrays. Those, together with thrusters, wiring and power conditioners weigh 300 kilograms. Payload, structure and propellant make up the rest of the stage.

Criteria for Performance Comparisons

Payload capacity and mission time have been adopted as provisional criteria fo merit for comparing propulsion-system and mission alternatives. Before discussing these applications, it should be pointed out that the term "payload" as used here is the gross payload; it includes guidance, control, telemetry and other such systems in addition to the science or engineering experiments. It does not include a separate electrical power supply in the solar-electric case, because for all but a few missions the power from the main panels is sufficient for housekeeping purposes during the powered flight and for data gathering and telemetry thereafter. This point is illustrated in figure 3 where the potential data transmission rate is plotted against the destination planet's solar distance. (The curves shown are based on antenna diameters of 210 and 20 feet at Earth and vehicle, respectively, and at the communications parameters as in reference 11.) The two solid curves are for solar power alone (10 kW at 1 AU; power varies with solar distance as determined in ref. 12); the dashed curves illustrate the effect of adding a 0.5 kW constant-power source (e.g., a Radioisotope Thermionic Generator or RTG) to the same solar-electric system. The upper curves apply when the destination point is at its closest approach to the Earth, while the lower ones apply at greatest separation.

For reference, the horizontal line at 48×10^6 bits/second represents real-time black and white commercial TV. This would correspond to a very ambitious experiment such as fine-resolution mapping of a planet's surface. Most experiments, however, can tolerate a data rate that is many orders of magnitude smaller than this. For example, the first Mariner/Mars vehicle had a minimum rate of 8 bits/second and required over a month to transmit its pictures back to Earth. The unaided solar-electric system could match this performance at Pluto's distance. If intermediate rates of 10^3 to 10^4 bps are considered more realistic, it appears that the unaided solar-electric system could support telemetry from Jupiter and possibly

Saturn. An auxiliary constant-power source such as an RTG would probably be required beyond Saturn.

On the other hand, payloads for chemical stages must, without exception, include a power supply that can handle the experiments, the telemetry and all the vehicle systems. Thus, it may be concluded that the SEP system's residual power represents a significant advantage in terms of communications capability except perhaps at Uranus, Neptune or Pluto. To match this capability (and assuming there is really a requirement for it), the ballistic system would need to be equipped with a separate 10 kW (at 1 AU) power supply. The mass of the separate power supply (presumably about 300 kg) would then have to be counted as a penalty against the ballistic system payloads for all missions except those to Uranus or beyond.

It should also be noted that, by the nature of the solar-electric stage, its vehicle subsystems such as guidance, attitude control and pointing, etc. must be designed to operate on a long term basis. Hence, they need not and should not be duplicated in the payload. For chemical stages, by contrast, the payload must be self-contained -- including its own attitude control and pointing system, midcourse guidance and propulsion, etc. These differences should be accounted for when cost comparisons between solar-electric and chemically-powered stages are attempted.

Synchronous Communication Satellite Mission

Because of the above mentioned points, the solar-electric system clearly has a natural payload advantage in comparison with chemical systems for missions which require substantial amounts of power -- such as the Synchronous Equatorial Communications Satellite Raising Mission. This mission has the objective of raising a television relay from a low-altitude initial orbit into a geostationary orbit. It is felt that the relay would have to weigh a minimum of 200 kilograms (not including the power supply) in order to perform any worthwhile purpose. In figure 4 is a comparison of the solar-electric and chemical systems' performance for this mission -- in terms of payload delivered as a function of transfer time required to raise the orbit, and the booster used.

Chemical systems require only a few days to accomplish the mission, and their performance is indicated by asterisks on the left for two boosters - Atlas/Centaur/Small Solid[1] and Titan IIID/Centaur. For each case two

[1] TE 364-4

payload levels are indicated. The higher one is the gross payload, which as mentioned before, must include a power supply. Since there is a requirement for 5 to 10 kW of power at the end of this mission, it is estimated that a 150 to 300 kg power supply is needed. Subtracting this from the chemical system's gross payload -- as indicated -- leaves a remainder which can more properly be compared to the solar-electric payload values.

In any case, the large chemical system can deliver very substantial payloads -- several thousand kilograms. The Atlas-based system delivers a considerably smaller payload, but it is still above the estimated minimum of 200 kilograms.

The solid curves indicate what could be done with the suggested standard solar-electric stage. Using a small, Thor-based launch vehicle, it would take 150 days to deliver the minimal 200 kilogram payload. Performance increases with transfer time, however, and it can match the 500 kilograms delivered by the Atlas/Centaur/Small Solid system in 280 days.

And, by using the same Atlas/Centaur booster, a payload increase of about 70 percent is obtained; i.e., payload increases from 500 to 850 kilograms, at a trip time of only 100 days. For this booster, SEP performance increases very rapidly with increasing transfer time, until at 500 days it has just about matched the performance of the Titan IIID chemical system.

Here it seems appropriate to dwell on two points which show up repeatedly in this discussion.

First, notice that for a given payload level, the solar-electric stage accepts a smaller launch vehicle. For example, at low payloads the Atlas/Centaur may be replaced by a TAT/Delta. For higher payloads the Atlas/Centaur could have replaced the Titan IIID/Centaur.

Second, note that with a given booster, it is always possible to get a large increase in payload by incorporating the solar-electric stage.

Costs will not be discussed in any detail in this paper, but several ways will be pointed out later in which these technical advantages might be translated into cost reductions.

This figure also shows the major disadvantage of the solar-electric stage: despite significant exceptions, its mission times do tend to be significantly longer than the all- chemical systems. In this example they were increased from a few days to hundreds of days. This is undesirable in itself, because of lower reliability, tying up the tracking network, and so forth. Moreover, in the case of a commercial satellite at least, the time increase also may imply a deferral of revenue -- which is an economic loss.

Area-Type Missions

On the other hand, there is a class of interesting missions for which the solar-electric stages' long propulsion times have a less serious effect. These are the so-called area-type missions in which the target is not a specific point, such as a planet, but is rather a general region of space. Examples include the Out-of-the-Ecliptic mission, the Solar Monitor, and the Close-Solar Probe. The reason why time is not such a major consideration for these missions is that useful data are being gathered all along the flight path. Therefore, even a premature system shutdown does not make the mission a complete failure; we could almost always salvage some worthwhile results.

The 1 AU out-of-ecliptic mission. - This has the objective of placing a payload in a solar orbit that is as highly inclined as possible to the Sun's equator -- in order to observe the Sun's high latitudes and polar regions. As in the previous case, it is felt that a 200 kilogram payload represents the practical minimum for a worthwhile mission.

In flight, the thrust is directed either straight up or straight down, relative to the instantaneous orbit plane as shown in figure 5. This sub-optimal but reasonably efficient steering program was chosen for its simplicity. The vehicle thus circles the Sun at a constant 1 AU while its orbit inclination gradually increases. The solar-electric power is alternately used for thrust in the nodal regions and for data gathering and telemetry in the antinode or maximum solar latitude regions. Mission performance is illustrated in figure 6 where gross payload is plotted against the final solar latitude for several boosters and mission times. Since several curves are shown, it is convenient to discuss them in groups. The two on the left are for the Atlas and large Titan-based all-chemical systems. Note their rapid fall off in performance -- even using a Titan IIID/Centaur/Small Solid it is only possible to reach 34 degrees with the minimal 200 kilogram payload. Mission time (time to reach the first antinode) here is 91 days.

The two short solid curves in the 30° - 50° region show the solar-electric stage's performance with the same two boosters. Performance has improved considerably. First, note that the solar-electric stage with Atlas/Centaur performs just about as well as the Titan IIID/Centaur/Small Solid system. In fact, it gets to 37 degrees rather than 34 at 200 kilograms payload. Secondly, when using the Titan IIID booster for both systems, the solar-electric stage now gives very good payloads in the 45 to 50 degree region -- a 50 percent higher final inclination than the all-chemical system can deliver. It must be conceded that mission time is now longer -- 465 days versus 91 -- but as previously mentioned, the system is gathering data at intervals all along the way. Thus the SEP stage's longer mission times are not necessarily an overwhelming disadvantage for this mission. On the contrary, a significant payload growth potential can be demonstrated by considering even longer mission times. For example, the third solid curve shows the performance available from the Titan IIID/Centaur/SEP combination at 3 1/2 years mission time. Payload capacities now range from over 1000 kilograms at 45° final inclination to 200 kilograms at 69° -- more than double the pdrformance of the 465-day case. But in fairness to the ballistic systems, it should be noted that they could use the Jupiter-swingby technique to great advantage at 3 1/2 years mission time. This gives a dramatic improvement -- as the right hand dashed curve shows. Even a 90 degree mission is possible, with very large payloads. But again, the solar-electric stage gives even higher performance (the upper solid curve) using the swingby mode. In general, the Jupiter-swingby out-of-ecliptic mission is so different from the 1 AU version -- and so much more complicated -- that we regard it as a complementary possibility rather than a competitor.

<u>The close solar probe mission</u>. - The other area-type mission to be discussed is the close solar probe. Here the object is to carry a payload (of 200 kilograms or more) as close as possible to the Sun's surface. Typical data for this mission are shown in figure 7. In part (a), the payload is plotted against the final perihelion radius for several alternative systems. If Atlas/Centaur is used with a small solid upper stage, it can carry about 200 kilograms inwards to 0.3 AU. This is the right hand dashed curve. From a scientific viewpoint, it would be desirable to get even closer to observe the Sun. Two techniques are illustrated. The Titan IIID/Centaur launch vehicle could be

used to send the same 200 kilogram payload in to 0.15 AU. Or, we could retain the Atlas/Centaur and use the solar-electric stage instead of the small solid. The right hand solid curve illustrates the performance of the 10 kW, 2600 second stage with 500 days trip time.

Note, firstly, that this system actually outperforms the much larger all-chemical system by getting the 200 kilogram payload in to 0.1 AU. Secondly, we could also combine the same solar-electric stage with the Titan IIID/Centaur and achieve still better performance -- down to solar impact for a small probe, as the left hand solid curve shows.

Thus, it is clear that a single solar-electric stage would have a substantial performance margin over a comparable all-chemical system, unless Jupiter-swingby trajectories are used (note the dashed curve at the upper left). In that case, even extending the electric trip time doesn't restore the SEP's advantage, as can be seen in figure 7(b). Here payload is plotted against trip time. Again, the Titan IIID/Centaur/Small Solid combination yields really impressive performance when a Jupiter swingby is used. Of course, we then must accept its 3 1/2 years trip time and other complications. And, although not illustrated here, the solar-electric stage does even better. Actually, the comparison is about as it was in the extra-ecliptic mission -- each trajectory profile has its own good and bad points and the corresponding missions are essentially different in nature.

In passing, it is worth noting that if minimum payloads are acceptable (200 kg) then the solar-electric stage can do a very close probe mission (e.g., .05 AU) in much less time than the large chemical system.

As previously mentioned, the area-type missions seem to be particularly well suited to an early application of a standard solar-electric spacecraft. This is because precision guidance is not required, simple steering programs can be used, and some useful data would be returned even if the propulsion system failed prematurely. The out-of-the-ecliptic mission has an additional advantage because the spacecraft remains at 1 AU and operates at constant power and heat flux, instead of the more general case where these parameters vary continuously.

Planetary Missions

Although area missions seem specially suitable for an early SEP application, there is a large variety of planetary missions for which the standard SEP stage approach also looks attractive. In general, we find the same sort of advantages we did for area missions. Namely, compared to an all-chemical system the standard SEP stage yields competitive performance with a smaller booster, or it yields significantly better performance with the same booster.

To illustrate the small-booster side of the argument, figure 8 shows the predicted performance for several planetary missions. Here the same fixed electric spacecraft launched by the Atlas/Centaur -- the solid curves -- is compared to the larger Titan IIID/Centaur/Small Solid combination -- the dashed curves. Gross payload is plotted against trip time. In the case of a Mercury orbiter or a Ceres rendezvous, the solar-electric system has a clear-cut advantage simply because the all-chemical system cannot do the mission at all. The two systems deliver about the same payload for the Neptune flyby. For Uranus, there is a small trip time advantage for the large all-chemical system and, for Saturn, there is a rather substantial advantage. Nevertheless, even in this case the performance of the solar-electric system is not unreasonable, especially if one remembers that we are comparing it to a much larger chemical system. The important point here is that a small Atlas/Centaur/SEP system can often deliver as much performance, or more (depending on the mission difficulty), than a much larger all-chemical vehicle. It has this advantage with a single design and for many missions and launch vehicles. In other words, it has an attractive multimission capability for planetary reconnaissance. It is also worth recalling that many of the easier planetary missions such as Mars and Venus orbiters are performed comparatively well with electric systems if a lot of power is required for surface mapping or high data return rate, since the power supply can be used for such purposes after the propulsion phase is over.

The second point concerns the comparison of SEP and chemical upper stage performance when the same launch vehicle is used. Recall that for the area-type missions we showed that the SEP stage on top of Titan IIID/Centaur would give major payload increases compared to a small solid. Figure 9 shows the same comparison for trips to the outer planets. Except for a change of scale, this is the same type of payload/trip time plot that was just mentioned. Now, however, the same large booster is used for both systems. The SEP stage clearly offers an appreciable performance advantage at Jupiter and a very substantial one for Saturn missions. It is overwhelmingly superior, both in terms of trip time and payload, for trips to Uranus or Neptune.

In fact, upon noting that the figure has logarithmic scales, it would appear that the solar-electric stage offers a revolutionary improvement in capability for outer planet flyby missions. For missions to Saturn and beyond, the performance seen here compares very favorably with what the ballistic system can accomplish by using a Jupiter swingby or "Grand Tour" approach. In the SEP case, moreover, launch opportunities occur once every 12 to 13 months, while Jupiter swingby and Grand Tour opportunities for ballistic outer planet trips have synodic periods of 13 to 179 years. Also it is true in this case, as it was for the area missions, that the solar-electric stage can also fly the Jupiter swingby - Grand Tour type missions, and shows a payload improvement when it does. This is because the first leg of a Jupiter swingby trajectory (to anywhere) always seems to involve a 500 to 700-day Earth-to-Jupiter travel time. And, as can be seen, the solar-electric stage can easily duplicate that trajectory and deliver a larger payload at the same time.

Growth Potential

Up to this point, the results presented have been based on a specific powerplant mass of 30 kg/kW and other parameters which represent current technology. Looking to the future, however, it is reasonable to expect the "state of the art" to improve. Perhaps the most dramatic change would be a substantial decrease in the weight of the powerplant. On figure 10 is shown a payload growth curve in terms of the overall propulsion system specific mass for a 40 degree out-of-the ecliptic mission using an Atlas/Centaur. With the current α of 30 kilograms per kilowatt it could deliver about 150 kilograms. The two curves show how payload increases as technology improvements reduce the specific mass, in one case for a fixed design and in the other case for a continuously reoptimized design. The limit on the left hand side is for bare silicon solar cells, which presently weigh about 5 kg/kW. Suppose for example, that the propulsion system specific weight was decreased from 30 to 15 kg/kW. This would at least double the payload - more than double it if the power and specific impulse are reoptimized.

Cost Savings

This concludes the discussion of mission performance. In retrospect, two trends have emerged again and again. For one, the solar-electric stage on top of Atlas/Centaur can essentially match the performance of a small solid stage[2] on the Titan IIID/Centaur booster.

For another, the standard solar-electric stage gives substantial, sometimes overwhelming advantages in terms of payload and/or trip times when the same booster is used.

This paper does not include an economic analysis, but it seems fair to mention that there may be some cost advantage under both of the above points. For one, we should be saving at least $13 million per mission in launch costs when we replace Titan by Atlas based launch vehicles, and perhaps $8 million per mission when a Thor based vehicle replaces Atlas. This is probably more than the solar-electric stage would cost in high-rate production.[3] Thus we would expect some net savings from the tradeoff, and this will help underwrite the development program for the solar-electric stage.

There are several ways to demonstrate a savings from the increased payload capacity of a given booster. First, if the mission has commercial applications, one can equate a larger payload with increased revenues. Or, the larger payload could mean a greater scientific return per mission. Secondly, the time may come when only one single, general purpose and presumably low cost booster vehicle is available for all users. For example, there is the Space Shuttle, which, when equipped with a Centaur-type of upper stage, would give roughly the same performance as was shown for the Titan IIID/Centaur. In this case, to get to the really large payloads, one must compare the solar-electric stage plus one Shuttle/Centaur launch with multiple Shuttle/Centaur/Solid Stage launches. Specific cost numbers will not be mentioned in this case, even as an example, because there are still many uncertainties in the Shuttle program. It does not seem unreasonable, however, to anticipate a net savings in this case also, when comparing the cost of the solar electric stage against the cost of at least one extra shuttle launch, plus at least one extra Centaur-type upper stage.

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[2]The small solid upper stage (TE 364-4) is used for comparison purposes herein because it has presumably been found to be more cost-effective than a small cryogenic upper stage (e.g., a scaled-down Centaur).

[3]Excluding payload-type items which would be present on both chemical and electrically-powered stages.

PROPULSION TECHNOLOGY

The current propulsion hardware status is reviewed in this section under the two functional headings: Power Supply Subsystem and Thruster Subsystem. The major elements of these two subsystems are discussed with the intent of justifying the rough weight and efficiency assumptions used in the mission performance estimates just given and also to focus attention on propulsion problems peculiar to an SEP stage.

Power Supply Subsystem

A block diagram of the major elements of the power supply is given in figure 11. The main source of electric power is the solar array. An auxiliary power source is needed to supply power at the start of the mission (prior to solar panel development), during solar occultations, or whenever the spacecraft gets too far away from the Sun (or too close) for solar cells to be effective. The latter situation is depicted in figure 12 which shows the power against Sun-distance variation assumed for the preceding mission estimates. At very small and very large distances the power output from the solar array is negligible. A battery could provide auxiliary power for the simplest missions, but a radioisotope thermoelectric generator (RTG) would be needed for missions beyond Saturn and a thermionic or a thermoelectric generator for close solar probe missions.

The raw power coming from the solar array must be transformed by the power conditioners into a form suitable for the thrusters. Another power conditioner is required to meet the power demands of the communication system, thruster controls, science experiments, general housekeeping, and so forth. Although the figure 11 diagram does not show it, there is a possibility that the auxiliary power supply could be used to drive, say, a single small thruster. This might be desirable for missions requiring a substantial amount of RTG power only at the destination and available enroute for primary propulsion or midcourse steering.

Solar array. - The conventional Mariner-type arrays weigh about 50 kg/kW, which is too heavy for primary propulsion. In order to reduce the array weight, Boeing has designed a large foldout array (ref. 13) using lightweight beryllium technology that should be capable of 21 kg/kW over a power range from 5 to 50 kW. This technology has been demonstrated as feasible and is ready for initiation of flight prototype development. General Electric has recently demonstrated the feasibility of a rollout array that operates as a window shade with the solar cells mounted on a flexible membrane (ref. 14). It weighs about 15 kg/kW over the power range of 5 to 20 kW. Both of these arrays require about 100 ft^2/kW and use silicon solar cells. The demonstrations referred to here both involved the construction of laboratory prototype panel-sections in addition to design efforts.

Unfortunately, these solar cells degrade slowly with time due to ultraviolet radiation, and proton and micrometeorite bombardment. The present uncertainty in predicting such degradation for interplanetary missions results in a 15 to 20 percent size increase in the array. Hopefully we can refine our knowledge of the solar array degradation by observing its characteristics in flight. In this regard, the results of SERT II indicate that after six months of operation in Earth orbit environment (underneath the Van Allen belts) the degradation is 11 to 12 percent which is slightly less than that predicted.

One future possibility that has the potential of even lighter weight, lower cost, and less degradation is the use of thin-film cadmium-sulfide solar arrays. These could be built using the rollout scheme and would have somewhat simpler packaging problems. At the moment, however, the cadmium-sulfide solar cells have low reliability (due to electrical instabilities) and are less efficient than the silicon cells by a factor of 2 or 3.

Auxiliary power. - The sole use of batteries for the auxiliary power source is severly limited by their poor energy storage capability per unit of weight. For near-Earth missions this may be acceptable, but for far-out, long-duration missions it is not possible to recharge the batteries from the solar array and RTG's are necessary. Besides long operating life, RTG's can provide about ten times as much thermal power as electrical power. This could be an important factor for outward missions beyond the asteroid belt where effective thermal control is difficult to accomplish with solar power along. Exclusive use of RTG power for electrical and thermal requirements has been recently proposed for the deep space missions using ballistic spacecraft (ref. 15). Existing RTG's meeting all safety provisions weigh about 500 kg/kW (electrical). A multihundred watt RTG development program is currently underway that is expected to reduce this figure to around 270 kg/kW using high temperature heat sources (ref. 16). Such an advanced RTG could be available in the mid-1970's. Further gains might also be achieved by jettisoning the safety shields after the spacecraft is injected on an escape trajectory although the practicality of this approach has not as yet been established.

Thermionic converters or thermoelectric flat-plate generators might be required for close solar probe missions. At 0.1 AU the specific weights are estimated to be around 10 to 15 kg/kW for either system (ref. 16).

Power conditioning. - The main power conditioners convert the low d.c. voltage of the solar array into high d.c. and low a.c. voltages required by the thrusters. In addition, to avoid thruster complications they must provide nearly constant output voltages even though the array output voltage may change as much as a factor of 4 to 1. The varying supply voltage is due to the change in light intensity and temperature as the spacecraft moves away from or toward the Sun. This is shown by a typical set of operating curves in figure 13. Current power conditioners use modularized transistor technology that can only accommodate a 2:1 change in input voltage (ref. 17). Increasing this to 4:1 would incur either a large weight penalty or necessitate using a different input line regulator for different missions. Using thyristor switching elements instead of transistors would eliminate this limitation although this technology is at least one year behind current transistorized power conditioning technology.

Although 5000 hour life tests have been successfully completed on certain power conditioners (e.g., SERT II prototype), the testing time on the newer, lightweight designs is much less. The table below summarizes the current status of JPL power conditioners (ref. 17):

Power rating	2.5 kW
Input voltage range	2:1
Efficiency	0.90
Mass	6 kg/kW in 1970 5 kg/kW expected in 1971
Expected reliability	0.96 for 10 000 hours
Testing time	1300 hours including 10 000 recycles due to thruster arcing

Testing of auxiliary power conditioners that weigh 4 kilograms has just begun (ref. 17).

There has been much recent interest in the high-voltage array concept which has the potential of eliminating much of the main power conditioner. Feasibility studies (ref. 18) indicate that such an array could reduce the combined array and power conditioner weight as much as 20 percent. Tests of a high voltage array connected to a small thruster at Lewis proved that this combination is indeed quite workable, but there are still too many uncertainties (such as array-space plasma interaction) involved to consider this scheme as currently available technology.

Thrust Subsystem

<u>Thrusters</u>. - It is possible to use a single thruster for some relatively simple missions, but in general it is better to use a multiple thruster array to provide flexibility. The available power will vary markedly during the course of many missions due in part to degradation but mainly due to the decrease (increase) in solar flux as the spacecraft moves away from (toward) the Sun. As shown in figure 12 this variation could be as much as 10:1. This may preclude the use of a single thruster since thrusters may not be able to be throttled over such a range without prohibitive decreases in efficiency. Reference 17 suggests 3:1 as limiting. At the present time, however, very little testing on throttling capability has been done; it is suspected that significant difficulty may be encountered in this area. In any case it is likely that individual thrusters must be turned on and off occasionally to match the solar array power variation. This power matching method offers two advantages: (1) the required throttling range per thruster decreases inversely with the total number of thrusters; and (2) thruster-installation reliability may be expected to improve through redundancy and shorter lifetime requirements per average thruster. Long thruster lifetime is a prime requirement since propulsion times range from 2400 to somewhat greater than 10 000 hours,

The basic technology of mercury ion thrusters is in hand. This technology is ready to be incorporated into designs of thrusters for particular applications. The first SERT II thruster operated for 3785 hours in space when failure occurred. The failure is thought to have been caused by local grid erosion that resulted in a small piece of grid material becoming lodged between the two grids and causing a high voltage short circuit. It is expected that avoiding this problem on future flights will not be difficult and that lifetimes approaching 10 000 hours are nearly at hand. Much of the recent R&D effort has been directed at the single glass-coated grid thruster which has already demonstrated better efficiencies (refs. 19 and 20) than the conventional two-grid thrusters. In the past the single-grid type of thruster suffered from very short grid life. However, very recent tests have supported the preliminary conclusion that the usual test facilities have a very detrimental interaction with the single-grid thruster. Alternate grids, designed to circumvent the facility problem, have been tested with very encouraging results.

In addition to lifetime, the thruster efficiency is of utmost concern since any inefficiencies in this element cause proportionate increases in the entire propulsion system weight. The research done in this area (fig. 14) has resulted in substantial improvements in thruster efficiency, particularly in the low specific impulse range (2000 to 4000 seconds) where the optimum specific impulse usually lies. At 3000 seconds a SERT II technology thruster would have an efficiency slightly more than 50 percent. Present 2-grid thrusters could achieve about 63 percent at the same specific impulse and 1-grid thrusters about 70 percent.

The table below summarizes some of the characteristics of an existing single-grid thruster (ref. 20) considered to represent the current state-of-the-art.

Peak input power	2.5 kW
Size	30 cm diam, 16 cm length
Mass	5.5 kg
Projected lifetime	10 000 hours
Specific impulse	2850 seconds
Overall efficiency	0.69

Propellant feed and storage system. - Propellant feed system technology is quite adequate now. SERT II uses low pressure nitrogen and a rubber diaphragm to expel mercury from the propellant reservoir. The JPL system consists of titanium propellant spheres, neoprene bladders, and Freon to provide passive, low pressure feed without pressure regulators. This simple system weighs less than 3 percent of the propellant weight for 80 kilogram capacity tanks.

Thruster control. - To increase the probability of mission success it is necessary to provide some sort of propulsion system redundancy. The SERT II philosophy was to provide completely independent thruster-power conditioner packages. In this approach, if either a thruster or a power conditioner fails the combined package fails and enough thruster-power conditioner units must be carried along as spares to insure reasonable mission success probability. Another approach (advocated by JPL) is to permit interchangeable connections between pairs of these subsystems. A logic module is needed to actuate and monitor a switching matrix in this case. These weigh 10 kilograms in JPL tests but are expected to weigh 5 kilograms as flight prototypes. In either approach, switches are needed to turn individual units on and off and these weigh about 2 kilograms. Other electronic elements are required to control the thrust vector orientation and to control and monitor the complete propulsion system as a whole. A typical block diagram of a propulsion system complete with all the various controls is given in figure 15. This particular layout is for the Solar Electric Propulsion System Technology (SEPST) demonstration program underway at JPL (ref. 17).

Propulsion System Mass Summary

A mass breakdown of the complete propulsion system in JPL's SEPST program is given in the following table (ref. 17) for 2 1/2 kilowatt, 3500 second I_{sp} thrusters.

TABLE I. - SEPST III WEIGHT SUMMARY

	Present, kg	Future, kg	
	(3 thrusters, 2 PC units)	(3 thrusters, 2 PC units)	(5 thrusters, 4 PC units)
Thruster (20 cm)	15.3	13.5	21.5
TVC actuators***	8.8	7.4	11.8
TVC thruster array and translator	55.0	22.0	22.0
Power conditioner (2.75 kW)	36.0	29.0	58.0
Controller (CC&S)**	5.2*	5.2	5.2
Switching Logic***	3.7	3.0	3.0
Switches***	5.0	3.0	3.0
Flexible cabling	5.7	5.0	7.0
Flexible feedlines	1.0	1.0	1.0
Caging for launch***	3.0*	3.0	3.0
Miscellaneous cabling*** and fittings	2.0*	2.0	2.0
Total	140.7	94.1	140.5

* Estimates for flight hardware

** Half of total CC&S

*** Fixed mass

Estimates of future flight-type designs are also included in the study. The major weight difference between the existing hardware (described in the table above) and the expected flight-type hardware is in the thrust vector control (TVC) structure and translator mechanism. Here a 60 percent weight reduction is anticipated. Since this is presently the heaviest element of the system, the expected overall 33 percent weight reduction (for 3 thruster systems) depends critically on this one element.

The solar array is not included in this breakdown since the ground-based SEPST program uses commercial electric power to simulate this large and expensive element. The solar array specific weight is expected to be nearly independent of power level with a value of about 15 kg/kW which should be increased by about 18 percent to account for degradation.

The masses of several of the elements listed in previous table depend on power level, specific impulse, and the number of thrusters. Therefore relationships have been worked out to indicate how the complete propulsion system mass varies with these variables. Figure 16 presents the current estimates of total propulsion system mass for flight-type hardware as a function of these variables (ref. 17). The specific mass ranges between 29 and 35 kg/kW at the 10 kilowatt power level depending on the specific impulse and the number of thrusters.

CONCLUDING REMARKS

Propulsion Technology

Most of the propulsion elements required for a solar-electric stage could be built with current laboratory technology within acceptable weight and efficiency limits. Still required is endurance testing of a complete prototype system under realistic environmental conditions. The SEPST program at JPL is continuing in order to demonstrate lifetime capability of many of these elements and NASA-Lewis is, of course, continuing to pursue thruster improvement. Although further development and qualification of many elements is definitely needed, no technical problems are apparent at this time that might preclude the successful demonstration of flight-rated hardware by the latter 1970's.

Power and Specific Impulse

As pointed out initially, a strong a priori case exists for choosing 10 to 20 kW of power and a specific impulse of 2500 to 4000 seconds as design points for a multipurpose solar-electric stage. The specific values of 10 kW and 2600 seconds, initially suggested as the result of prior Lewis and outside work, have been shown to be reasonable in this study. Performance is satisfactory for a substantial range of missions and does not depend critically upon the precise power and specific impulse used. The values mentioned should be interpreted as a first cut, however, rather than as a "final" or "optimum" choice. Clearly further study is warranted, to confirm or refine these values. The ultimate result will probably be more sensitive to policy considerations (e.g., mission priorities) than to numerical refinements.

Mission Performance

The suggested 10 kilowatt, 2600 seconds SEP stage has been evaluated -- in terms of payload capacity and mission time -- for missions ranging from near-Earth to a Neptune flyby. Launch vehicles ranging from TAT-Delta to Titan IIID-Centaur were used. In general, this fixed-propulsion-system stage was found to have excellent multimission performance. In comparison with an efficient solid chemical upper stage (TE 364-4), the SEP stage normally provides either (1) equal payload when using a smaller launch vehicle, or (2) significantly greater payloads when the same launch vehicle is used. In fact, it can perform certain very difficult missions such as the Mercury orbiter or asteroid rendezvous which are virtually impossible for a chemical system. Although the relative-cost comparison is not clear at this time, it is not obviously discouraging. The SEP upper stage will undoubtedly cost more to develop and manufacture than one based on a small solid motor. This, however, is offset to some degree by the SEP stage's ability to use fewer or smaller launch vehicles, or fewer launches of a standardized launch vehicle such as the shuttle.

From the performance viewpoint, the major disadvantage of the SEP stage is that its mission times tend to be uncomfortably long -- in many cases, significantly longer than an alternative all-chemical system would require. It should be recalled, however, that area missions (Solar Probe, Solar Monitor, and Extra-Ecliptic) produce useful data more or less continuously during their mission times. Thus, a premature system shutdown would not represent a total mission failure. Moreover these missions, and also the Mercury orbiter and asteroid rendezvous, are very difficult and would require a larger and more elaborate chemical system than was considered here. For most outer-planet missions, the SEP stage actually would yield the lower mission time.

It should be recognized that the performance shown here for both the SEP and chemical stages is based upon numerous simplifications. On the other hand, current studies being performed by others tend to confirm the present results. Further calculations, based upon more realistic solar-system, trajectory, vehicle and subsystem models, are definitely needed to confirm and refine the present data.

Preferred Missions and Launch Vehicles

Two launch vehicles, Atlas/Centaur/SEP and Titan IIID/Centaur/SEP, have been shown able to accommodate a wide range of mission objectives, flight times, and payload levels. There is no clear requirement for an intermediate sized launch vehicle, and there is only one mission (the synchronous satellite raiser) for which a smaller vehicle might be useful. It is therefore concluded that the SEP stage should be physically and functionally compatible with the Centaur stage. Launch-environment and loads criteria should reflect both the Atlas and Titan IIID boosters.

The selection or ranking of missions is essentially a process of judgment and will depend as much upon "priority" or fiscal considerations as upon technical ones. It should be noted, however, that the multimission vehicle capability can be built up incrementally if this were desirable for technical or developmental reasons. This would impose a definite, but apparently reasonable, sequence of performing missions. This sequence, in order of increasing complexity, is given in Table II below. Missions are listed by name in the left hand column with the appropriate launch vehicle indicated by footnotes; the next six columns list factors that have significant design implications; and the two right hand columns identify the required version of the SEP stage and its characteristic design features.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio February 26, 1971
124-08-41

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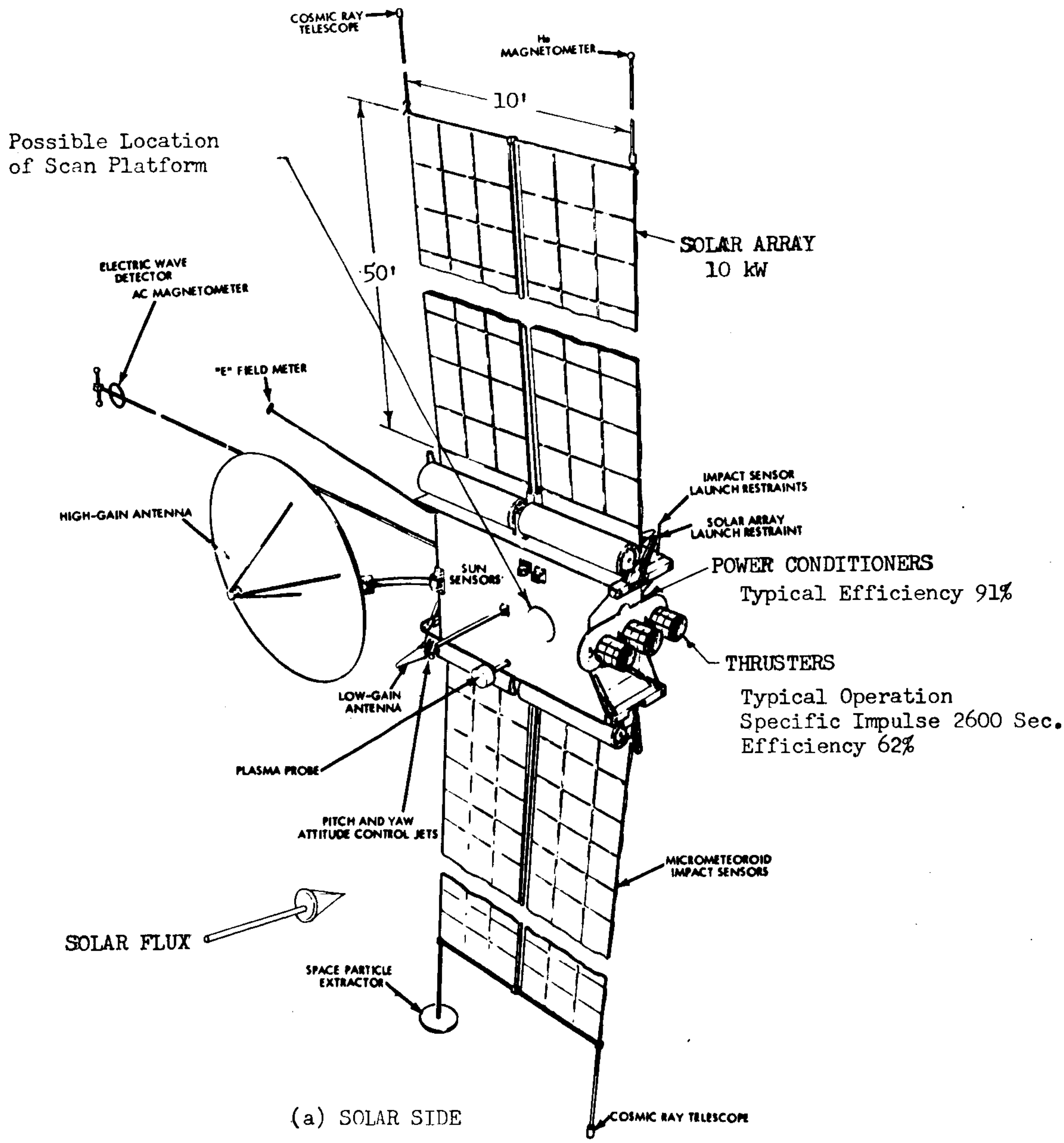
TABLE II. - PREFERRED MISSIONS FOR SEP STAGE

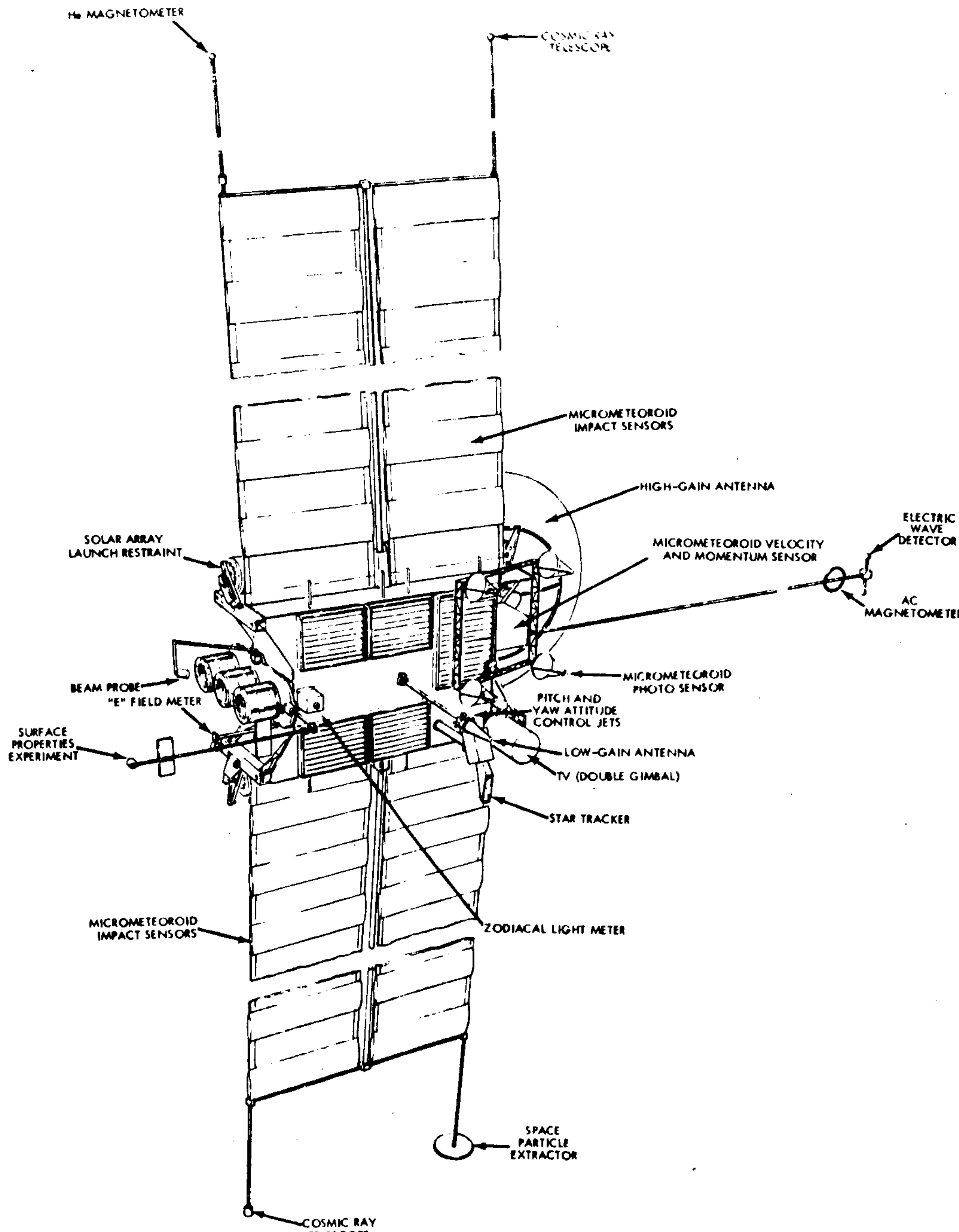
Missions	Power Profile	Solar Orientation	Thermal Environment	Navigation Requirement	Communication Distance	Mission Time, Days (approx)	SEP Version	Characteristic Design Features
1 AU, Extra-Ecliptic; Solar Monitor. (a) or (b)	Constant	Constant	Constant; moderate	area	under 0.5 AU	465	I	Basic vehicle, Nearly constant power, fixed panels, fixed engines, passive thermal control, autopilot plus Earth-radio guidance
Synchronous Communications Satellite. (a)	Constant (c)	Variable	Constant except for Earth-shadow passages; moderate	area	25,000 miles	100-300	II	I plus swiveling solar panels
Close Solar Probe. (a) or (b)	Variable	Variable	Variable; hot	area	1 AU (approx)	up to 500	III_a	I plus active thermal control and variable power and throttling capability for hot (inwards) missions
Asteroid Belt Fly Through. (a)	Variable	Variable	Variable; cool	area	2-4 AU	up to 700	IV_a	I plus active thermal control and variable power and throttling capability for cool (outwards) missions
Mercury Orbiter. (a)	Variable	Variable	Variable; hot	precision	1-2 AU	500	III_b	III_a plus precision guidance, small chemical retro motor
Asteroid Rendezvous. (a)	Variable	Variable	Variable; cool	precision	2-4 AU	600-700	IV_b	IV_a plus precision guidance, small chemical retro motor
Major Planet Flyby. (d), (b)	Variable	Variable	Variable; cool	precision	5-20 AU	600-3000	V_a	IV_a plus precision guidance, high powered telemetry, RTG auxiliary power supply
Major Planet Capture. (b), (e)	Variable	Variable	Variable; cool	precision	5-20 AU	600-3000	V_b	V_a plus small chemical retro motor

Notes: (a) Atlas-Centaur Launch
(b) Titan IIID-Centaur Launch
(c) Decreases slowly due to Van Allen proton degradation
(d) Followed by solar system escape
(e) High elliptic parking orbit



(a) SOLAR SIDE

Figure 1.– Solar Electric Multi-Mission Spacecraft ;
Typical Configuration (ref. 7)



(b) ANTISOLAR SIDE

Figure 1. - Concluded

MISSION A: 150 DAY SYNC. SAT. RAISER - ATLAS/CENTAUR

B: 700 DAY CERES RENDEZVOUS - ATLAS/CENTAUR

C: 500 DAY 0.1 AU SOLAR PROBE - TITAN III/CENTAUR

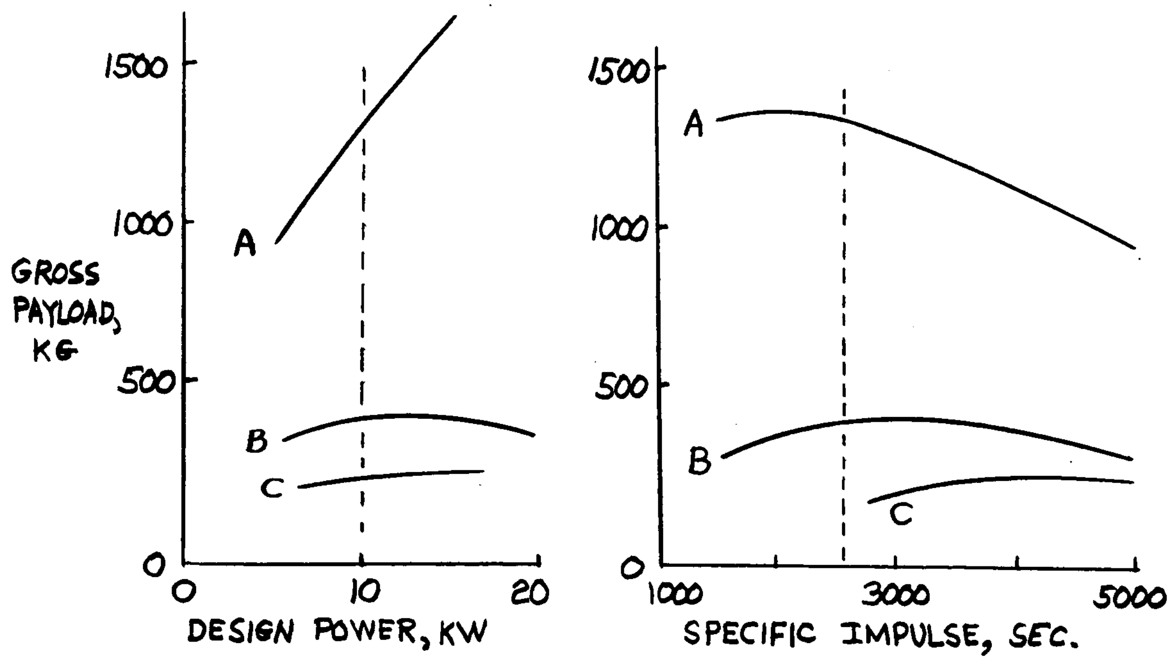


FIGURE 2.- SELECTING SOLAR ELECTRIC STAGE DESIGN VARIABLES

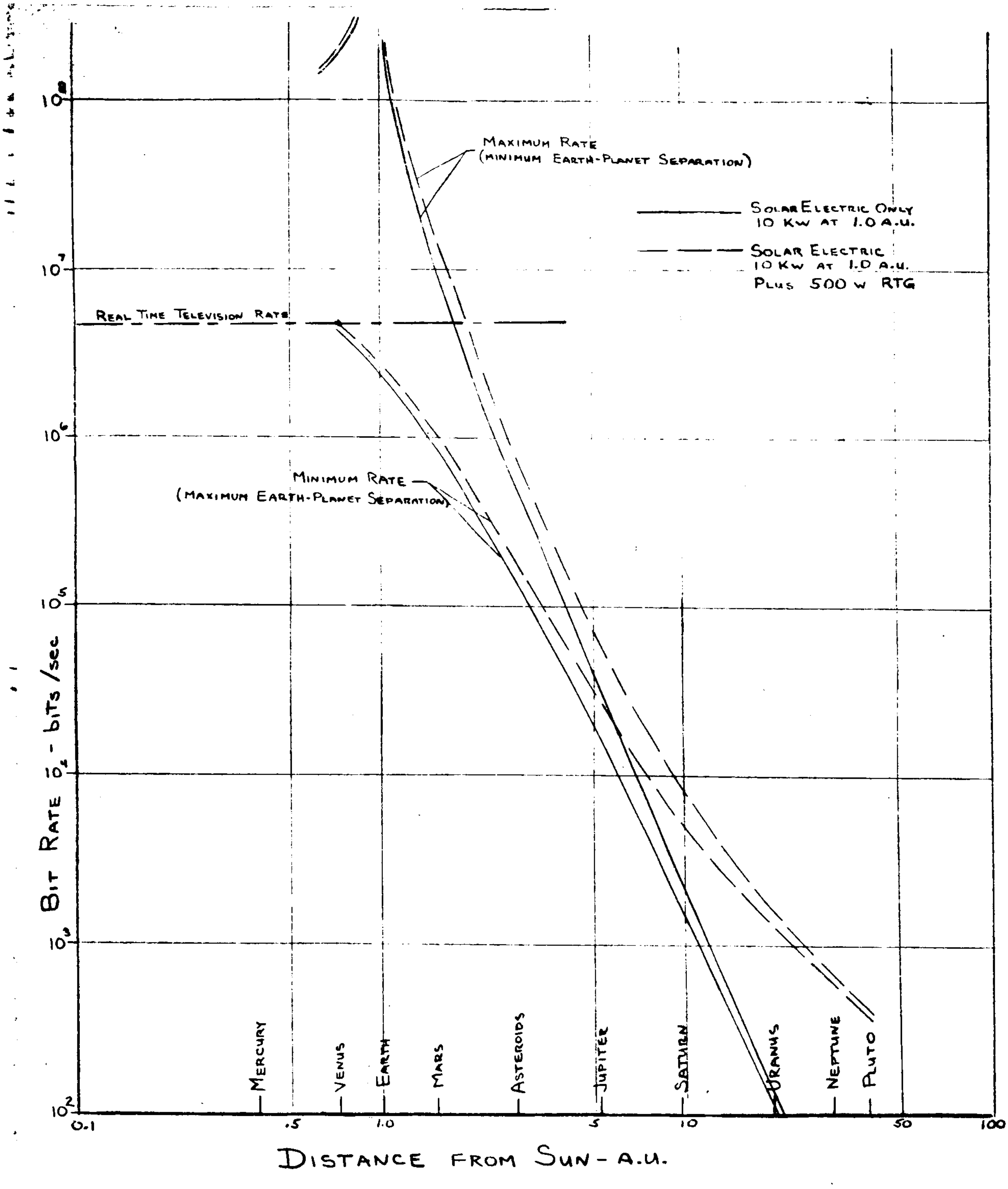


Figure 3: Communications Capability of Solar-Electric Spacecraft

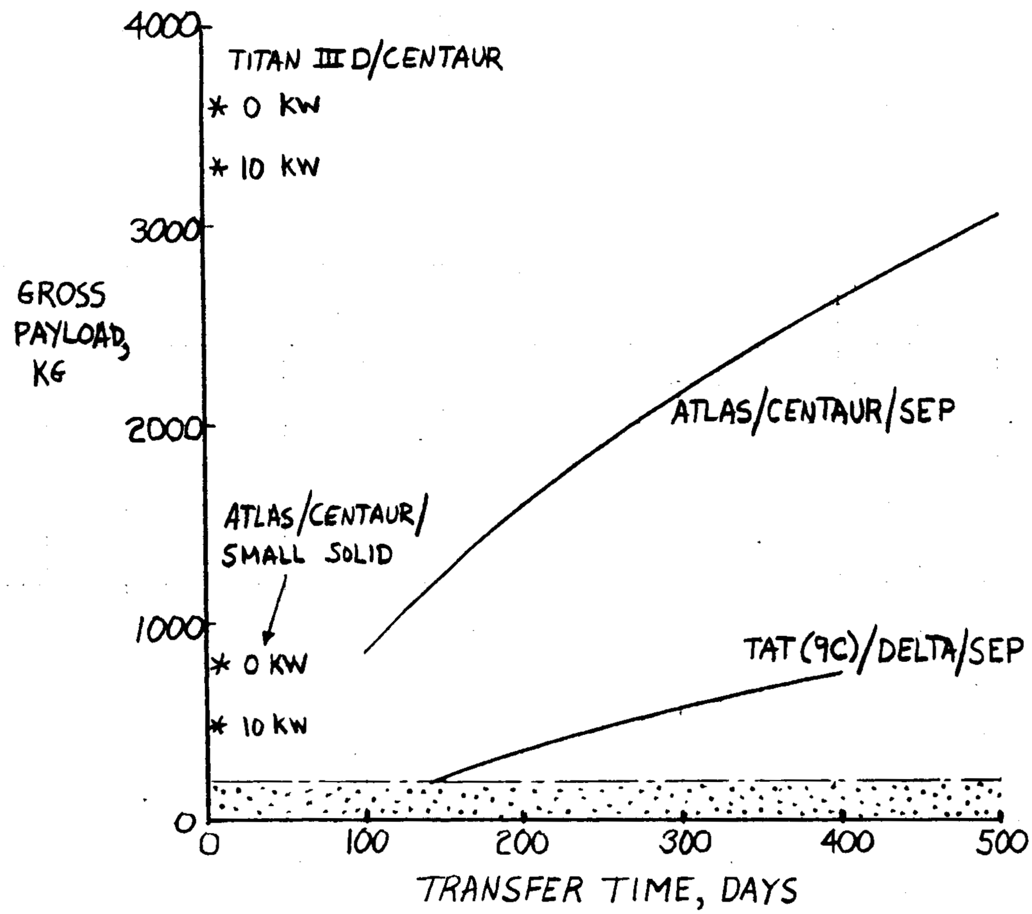


FIGURE 4. SYNCHRONOUS SATELLITE RAISING MISSION

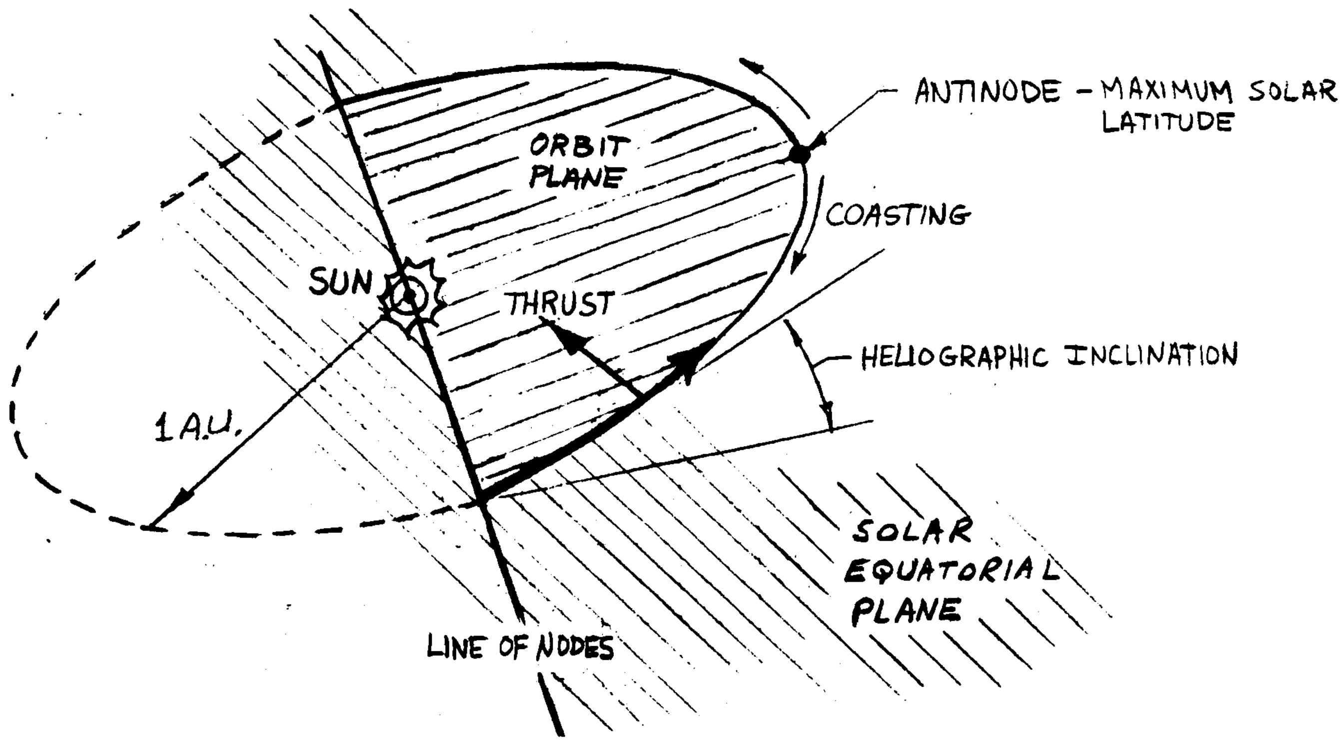


FIGURE 5.- TRAJECTORY PROFILE FOR 1.0 AU EXTRA-ECLIPTIC MISSION

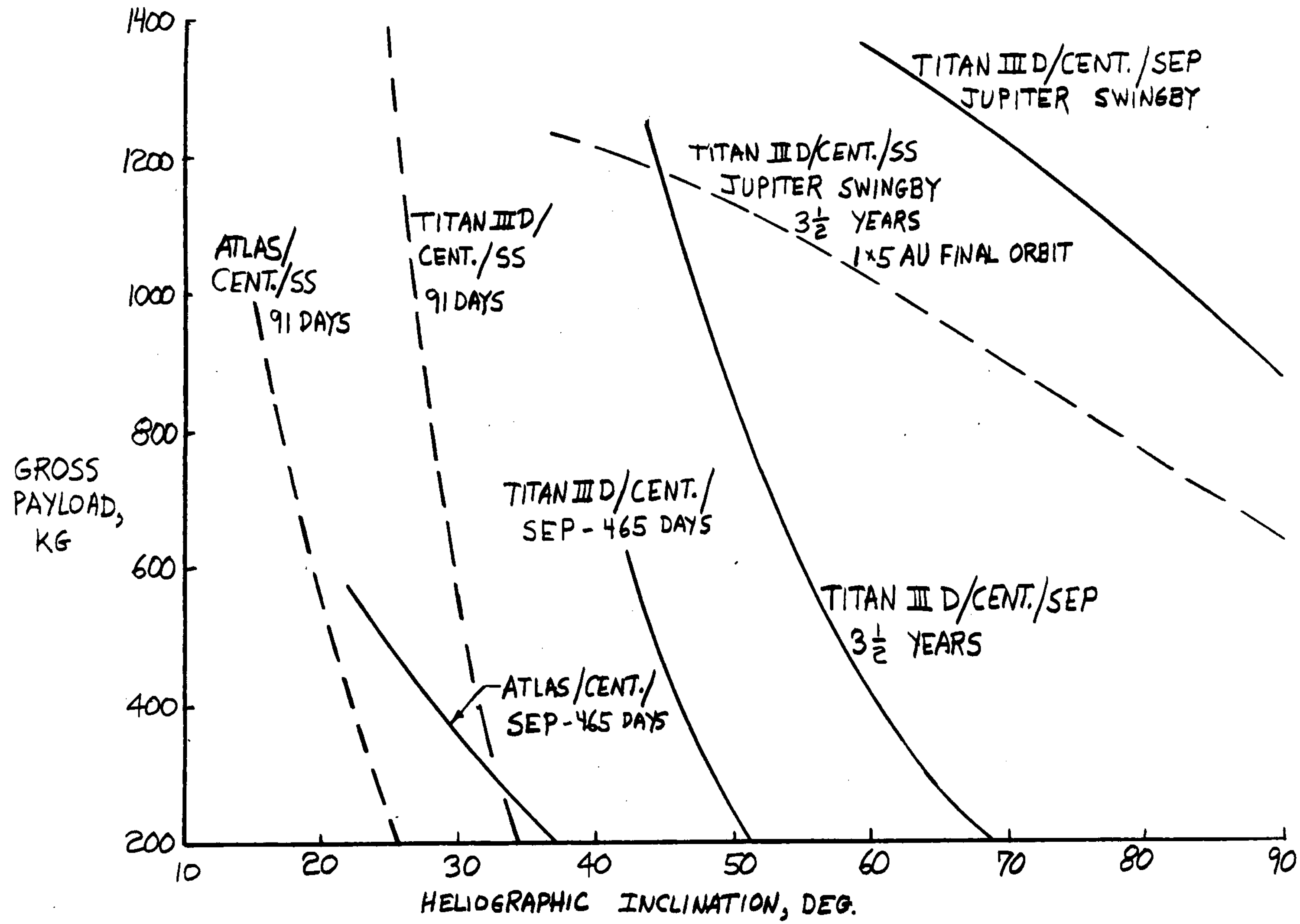


FIGURE 6. 1.0 AU EXTRA-ECLIPTIC MISSION

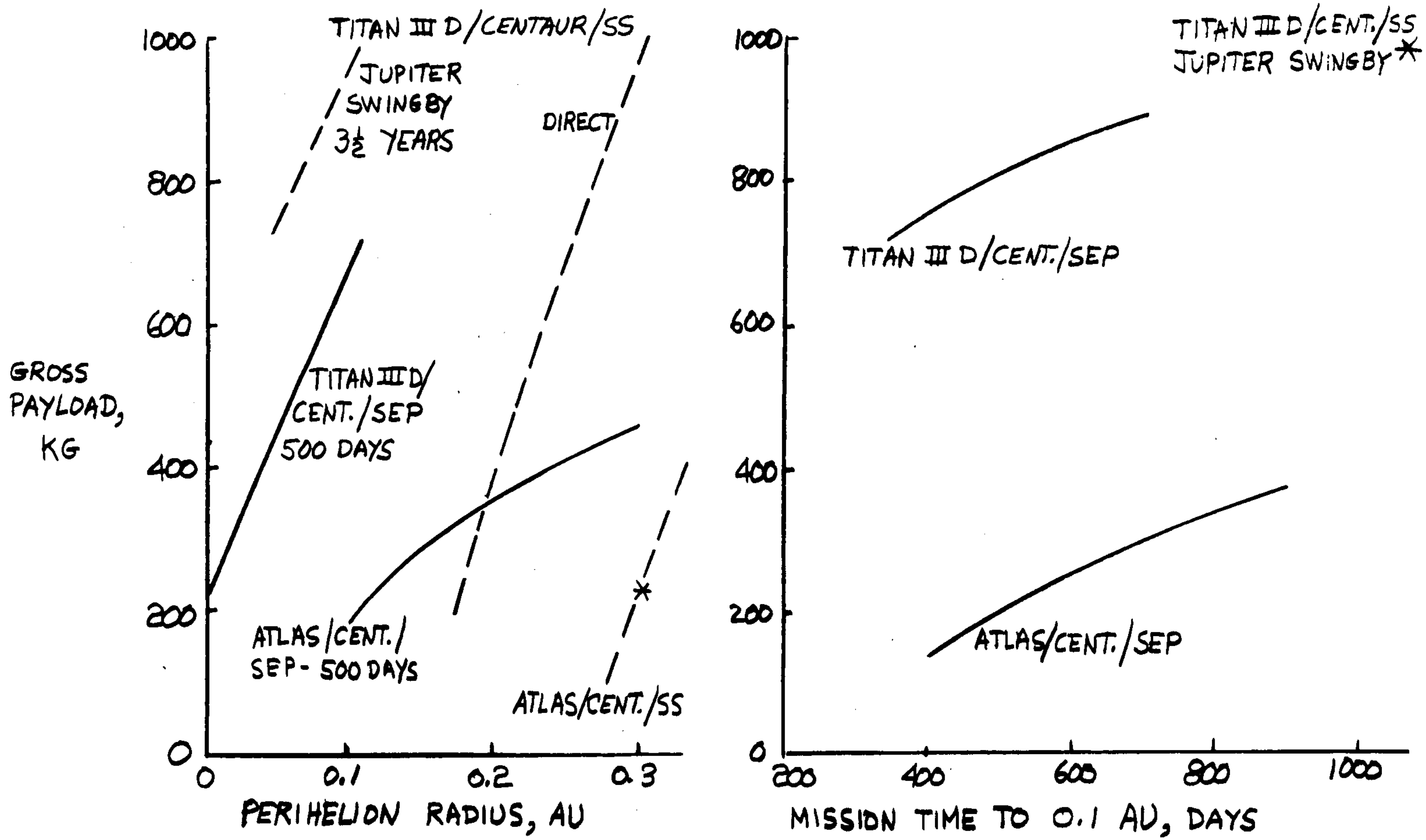


FIGURE 7. - CLOSE SOLAR PROBE MISSIONS

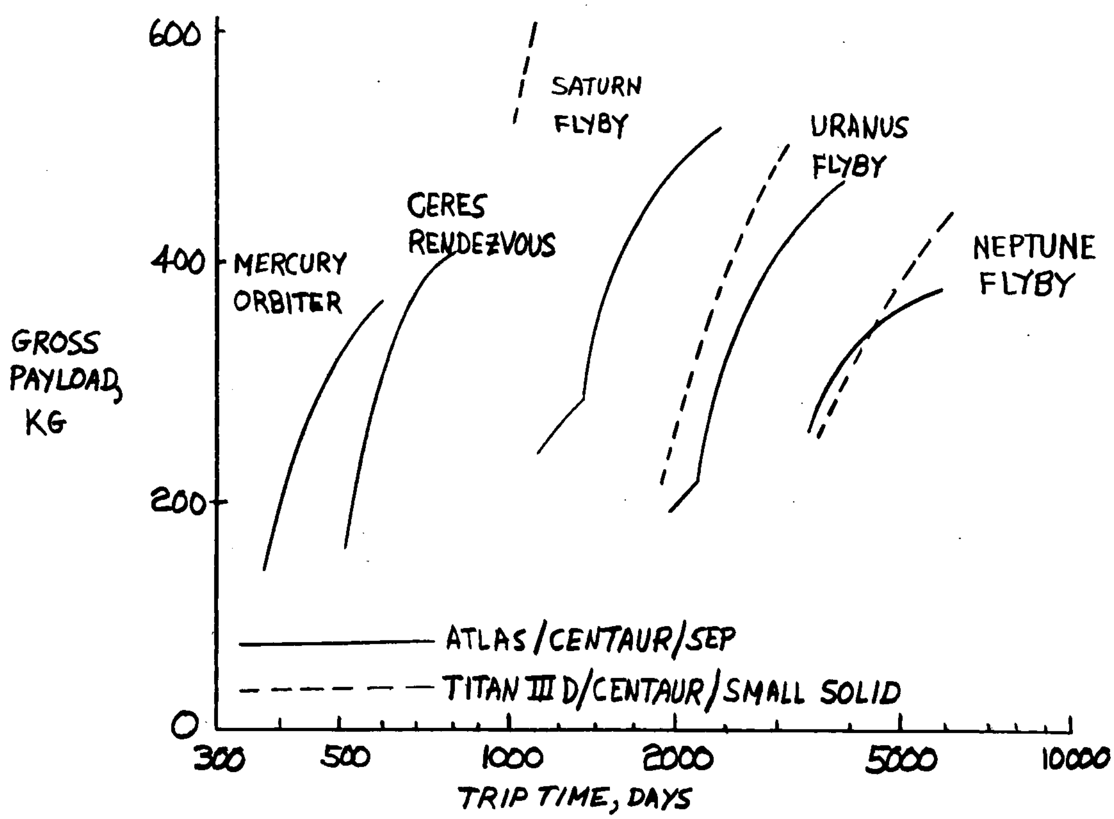


FIGURE 8- PLANETARY MISSIONS

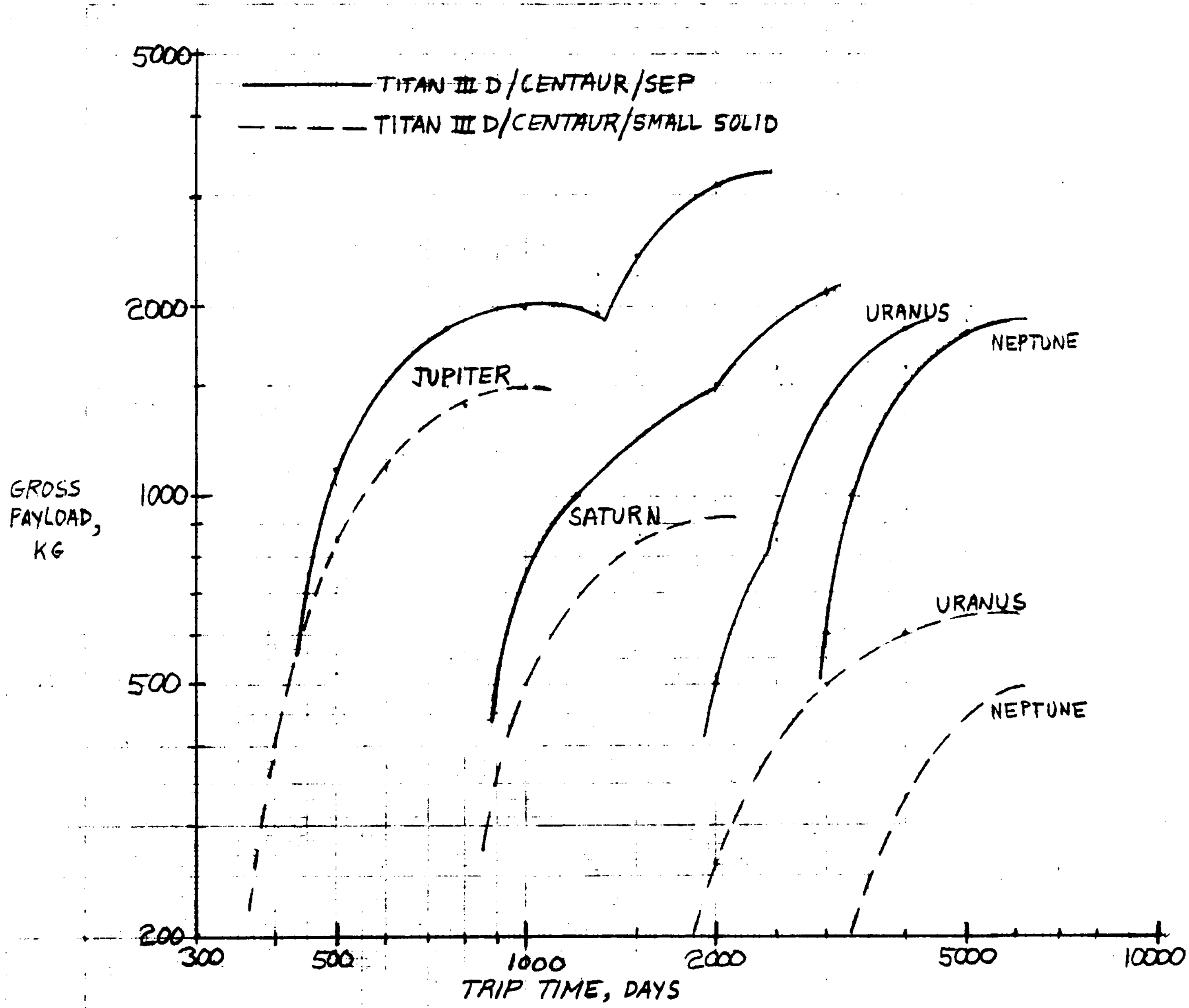


FIGURE 9 - PLANETARY MISSIONS

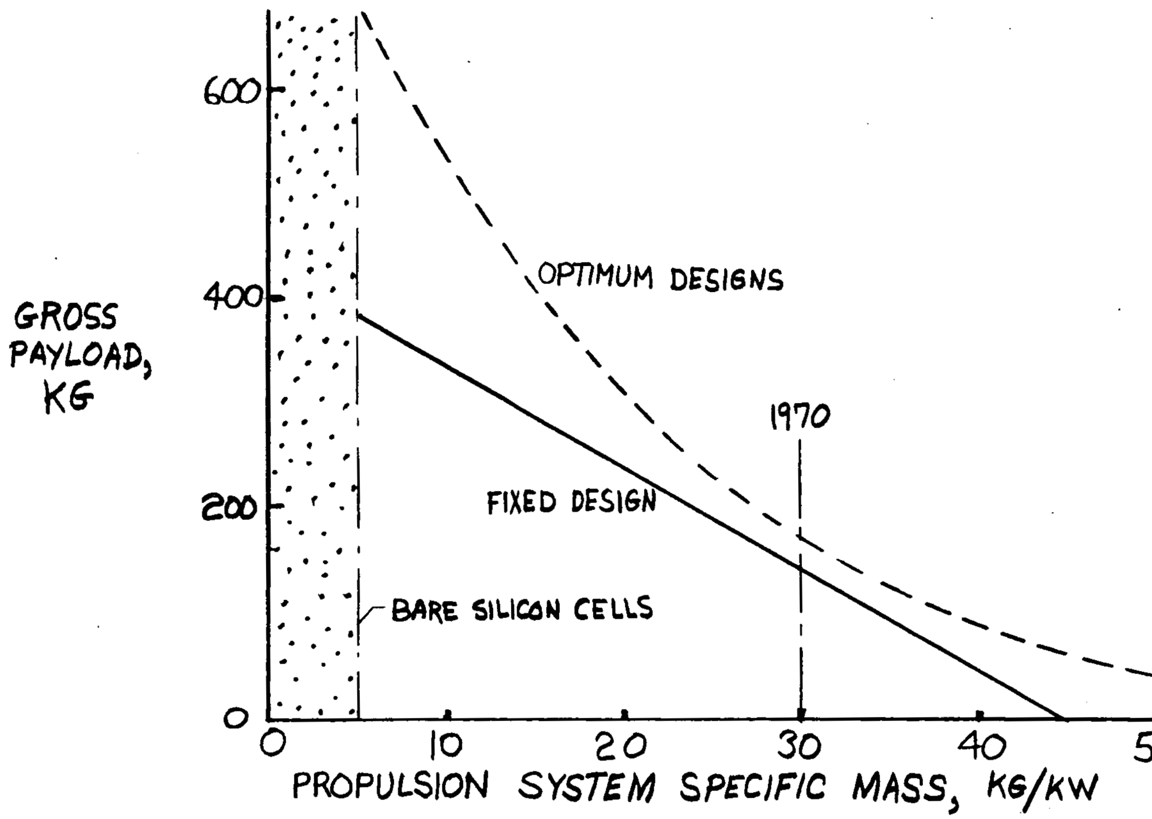


FIGURE 10. - 40° EXTRA-ECLIPTIC MISSION. ATLAS/CENTAUR/SEP, 465 DAYS.

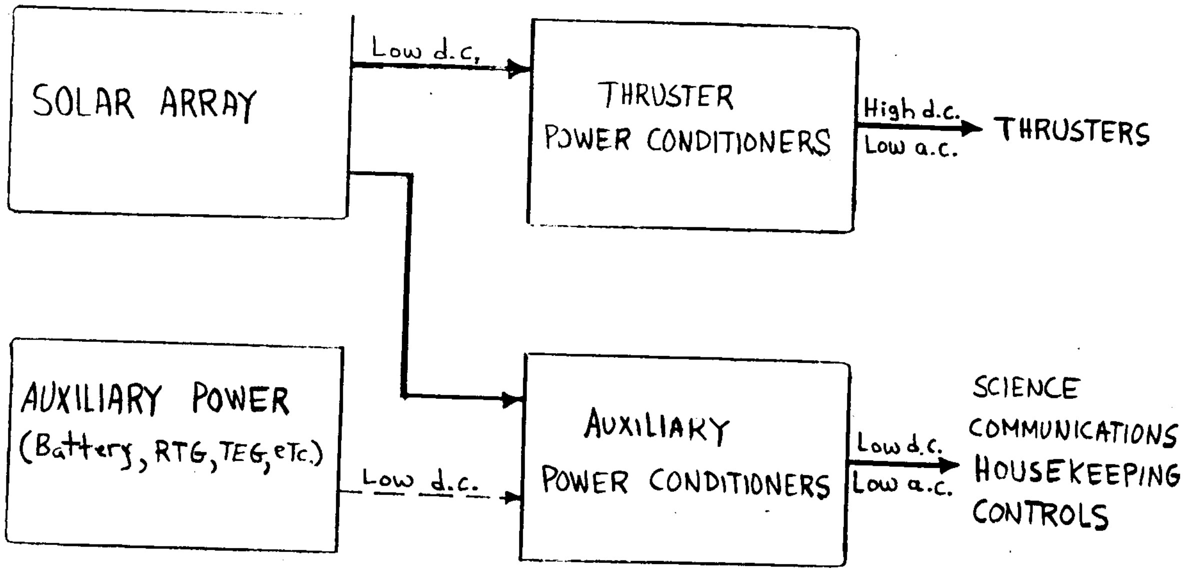


FIGURE 11.- POWER SUPPLY SUBSYSTEM BLOCK DIAGRAM

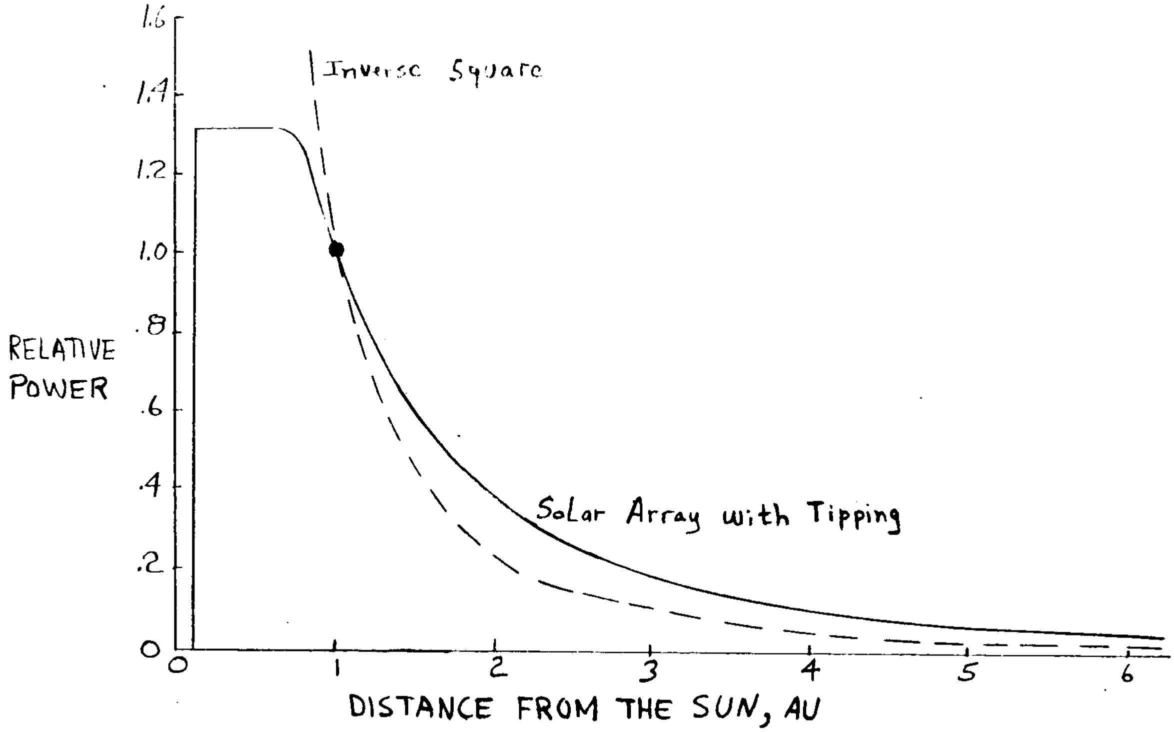


Figure 12 - SOLAR ARRAY POWER VARIATION

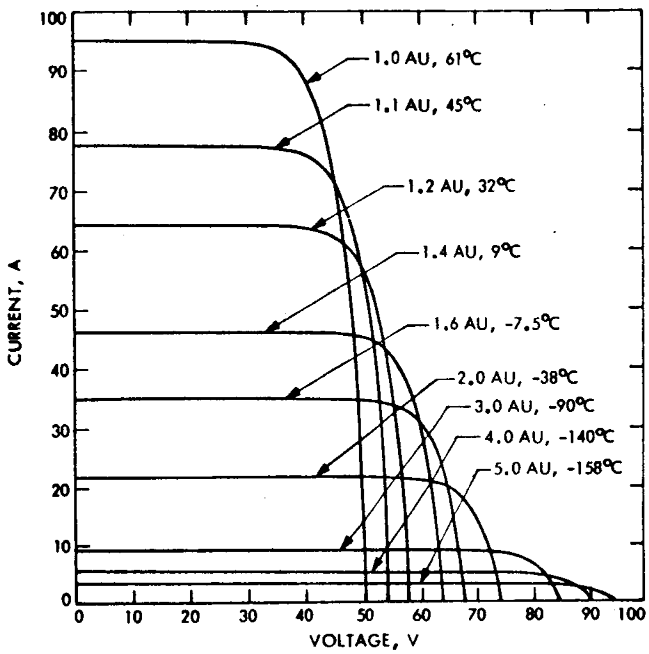


Figure 13. Solar Array Current vs Voltage Characteristic (Typical)

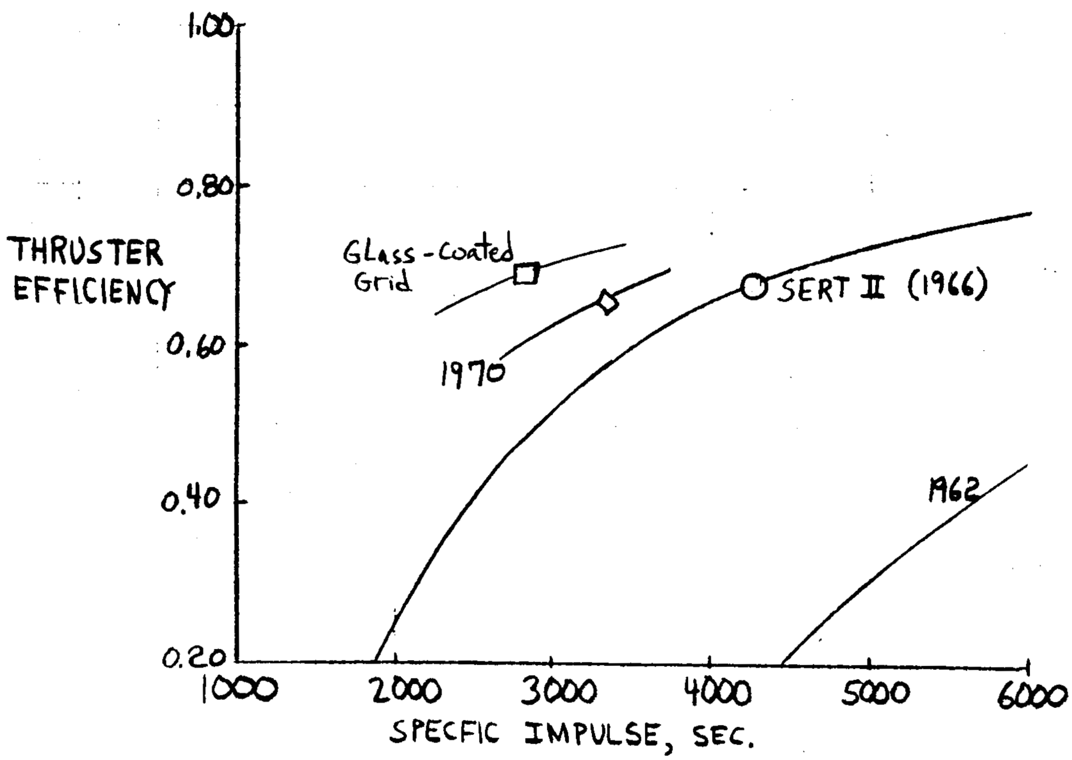


FIGURE 14- THRUSTER PERFORMANCE

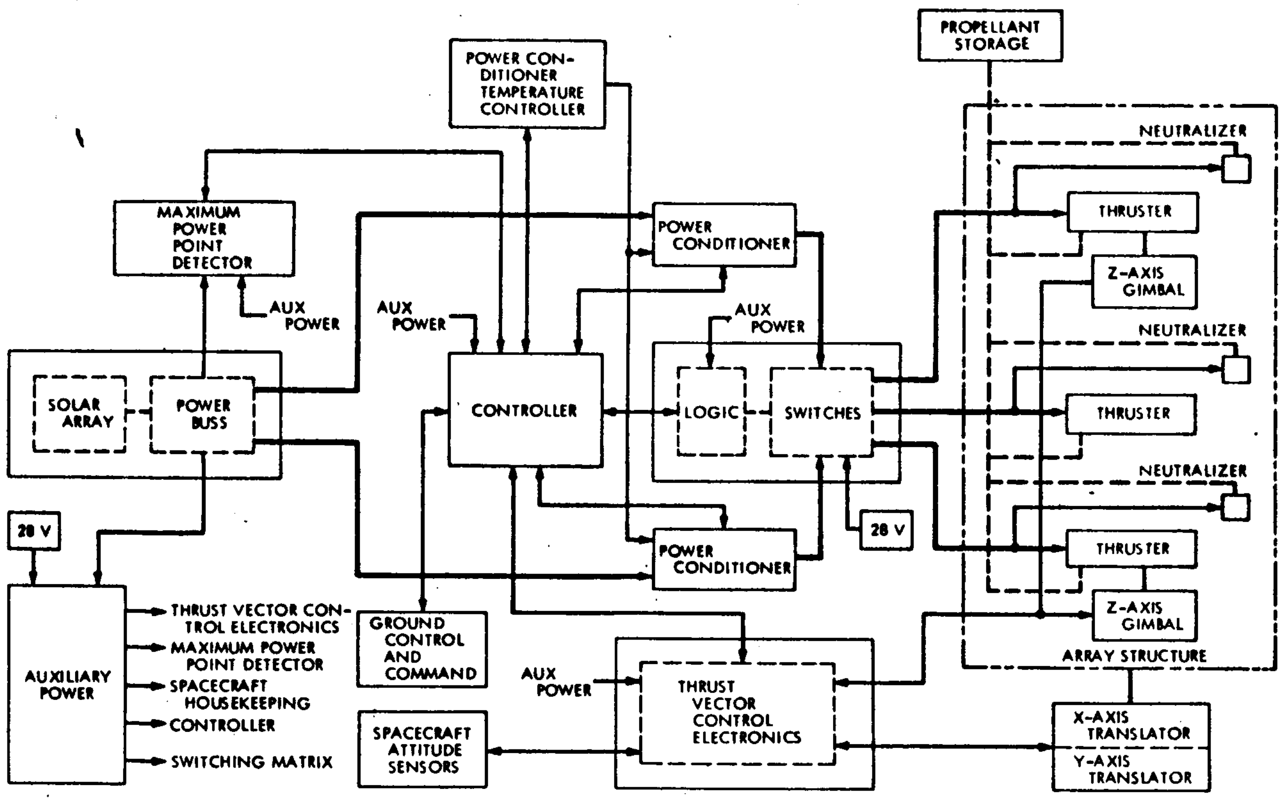


Figure 15. Propulsion System Block Diagram

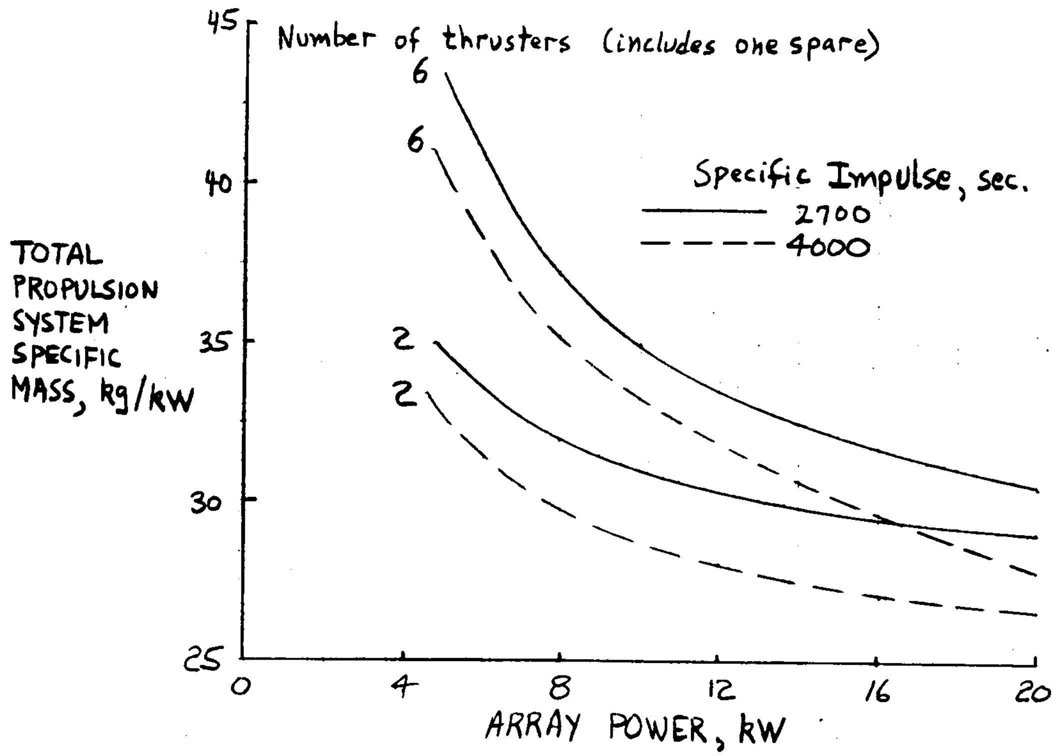


FIGURE 16. TOTAL PROPULSION SYSTEM MASS.